50

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Space Programs Summary 37-42, Vol. VI

Space Exploration Programs and Space Sciences

For the Period September 1 to October 31, 1966

JET PROPULSION LABORATORY

CALIFORNIA INSTITUTE OF TECHNOLOGY

PASADENA, CALIFORNIA

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Preface

The Space Programs Summary is a six-volume bimonthly publication designed to report on JPL space exploration programs and related supporting research and advanced development projects. The titles of all volumes of the Space Programs Summary are:

Vol. I. *The Lunar Program* (Confidential)

Vol. II. *The Planetary–Interplanetary Program* (Confidential)

Vol. III. *The Deep Space Network* (Unclassified)

Vol. IV. *Supporting Research and Advanced Development* (Unclassified)

Vol. V. *Supporting Research and Advanced Development* (Confidential)

Vol. VI. *Space Exploration Programs and Space Sciences* (Unclassified)

The Space Programs Summary, Vol. VI, consists of: an unclassified digest of appropriate material from Vols. I, II, and III; an original presentation of the JPL quality assurance and reliability efforts, and the environmental- and dynamic-testing facility-development activities; and a reprint of the space science instrumentation studies of Vols. I and II. This instrumentation work is conducted by the JPL Space Sciences Division and also by individuals of various colleges, universities, and other organizations. All such projects are supported by the Laboratory and are concerned with the development of instruments for use in the NASA space flight programs.

Approved by:

W. H. Pickering, *Director*

Jet Propulsion Laboratory

Contents

THE LUNAR PROGRAM

THE PLANETARY–INTERPLANETARY PROGRAM

THE DEEP SPACE NETWORK

I. *Surveyor* Project

THE LUNAR PROGRAM

A. Introduction

The *Surveyor* flight spacecraft are designed to span the gap between the *Ranger* Project and the *Apollo* Project by making soft landings on the Moon to extend our knowledge of lunar conditions and determine the suitability of sites for proposed *Apollo* spacecraft landings. Hughes Aircraft Company (HAC), Space Systems Division, is under contract to fabricate the *Surveyor A-21* spacecraft. The launch vehicle, a combination *Atlas/Centaur*, is provided by General Dynamics/Convair. Control, command, and tracking functions for the *Surveyor* missions are performed by the JPL Deep Space Network and Mission Operations System.

Surveyor I, the first flight spacecraft, was launched from Cape Kennedy, Florida, on May 30, 1966, and soft-landed on the Moon on June 2, 1966. By June 14, when lunar sunset occurred, approximately 100,000 commands had been received by the spacecraft, and 10,338 pictures of the spacecraft and its immediate vicinity had been transmitted.

Surveyor II, the second flight spacecraft, was launched from Cape Kennedy on September 20, 1966. The spacecraft performed nominally until the command was given for midcourse thrust execution. At that time, one vernier engine did not ignite. The resulting imbalance of thrust from the other two vernier engines imposed a tumbling motion on the spacecraft, from which it failed to recover. Subsequent attempts to fire the vernier engines increased the tumbling rate. The main retrorocket was ignited and burned for approximately 30 sec. Contact was then lost with *Surveyor II*. Continuous receiver search for the spacecraft was unsuccessful, and the *Surveyor II* mission was therefore terminated on September 22. At approximately 03:18 GMT on September 23, the spacecraft impacted the lunar surface.

Data of considerable value were obtained from *Surveyor II* up to the point of the midcourse thrust maneuver. These transmitted flight data, as well as failure data, are undergoing analysis in order to preclude a similar failure in the upcoming Mission C, currently scheduled for the first quarter of 1967.

B. *Surveyor II* Mission Operations

1. Launch Through Sun Acquisition

The *Surveyor II* (SC-2) spacecraft was launched from Cape Kennedy, Florida, at 12:32:00 GMT on September 20, 1966. The booster engine was jettisoned at 12:34:28 and the nose fairing 56 sec later. *Atlas* sustainer-engine cutoff and vernier-engine cutoff occurred 10 sec prior to *Atlas/Centaur* separation at 12:36:08, and injection took place at 12:43:24. At 8 sec prior to spacecraft/*Centaur* separation, which occurred at 12:45:25, the spacecraft began a negative roll turn and rolled for 72 deg before the acquisition Sun sensor was illuminated. At 2 min, 16 sec after separation, the spacecraft stopped rolling and started a positive yaw turn. The spacecraft yawed 16 deg before the primary Sun sensor indicated lock-on to the Sun. Sun acquisition was complete at 12:48:38. Less than 2 min afterward, the solar panel elevated to the transit position, as indicated by the secondary Sun sensor cells.

2. Star Acquisition

The star mapping maneuver began approximately 5 hr, 45 min after Sun acquisition when cruise mode, manual delay mode, and positive-angle maneuver commands were sent to the spacecraft. At 18:37:32, the roll command was sent to start the actual positive-roll maneuver at a rate of 0.5 deg/sec, and the roll began 5 sec later. The telemetry subsystem was in a coast mode transmitting at 1100 bits/sec. Due to a small number of identifiable stars and variable-intensity indications from the Moon and Earth, it was decided to perform two complete revolutions for star mapping prior to acquiring Canopus. The first map was made using omnidirectional antenna B, and the second map and subsequent acquisition were made using omnidirectional antenna A.

It was expected that star-intensity signals would be increased 20% due to the installation of a Sun filter having 20% increased filtering action. During the mission, Canopus did not yield a usable lock-on signal, suggesting that the sensitivity of the intensity signal was increased more than the planned 20%. However, it is also possible that Canopus itself is brighter than predicted. Due to the lack of a Canopus lock-on signal, it was necessary to use a manual lock-on command to null a 2-deg roll error angle which resulted when the roll was stopped past the center of the field of view by a cruise-mode command. This latter command had been sent when Canopus was in the field of view during the third revolution.

3. Midcourse Correction Maneuver

Premidcourse maneuvers consisting of a positive 75.3-deg roll turn and positive 110.5-deg yaw turn were accomplished satisfactorily on September 21. Vernier engine ignition began at 05:00:02.5, but Engine 3 did not ignite. The spacecraft began to tumble, saturating the gyro error signals. After termination of midcourse thrusting and until the rate mode was commanded on, the pitch and yaw gyro error signals varied from positive to negative. After this, the signals returned to their original saturated positions. When the flight-control subsystem was returned to the inertial mode at 07:28:25 (first nonstandard vernier ignition), the gyro error signals did not indicate any change. Since it was believed the tumble rate might be small enough for the gas jet attitude-control system to dampen it out without using an excessive amount of nitrogen, the gas jet amplifiers had not been inhibited immediately after thrust termination. The postmidcourse tumble rate was indicated to have a period of about 13 sec. However, later analysis showed a maximum rate of approximately 420 deg/sec, resulting in a period of less than 1 sec. When it became obvious that the nitrogen system would not be able to remove the angular rates, the gas jet amplifiers were inhibited at 05:14:29 with an estimated 2.16 lb of the original 4.50-lb nitrogen supply remaining and a spacecraft tumble rate of approximately 300 deg/sec.

During two subsequent attempts to ignite Engine 3, Engines 1 and 2 ignited normally, but no indication was received that Engine 3 had ignited. The vernier engines were later commanded on for 0.2 sec five times in succession every 5 min. Each time, a 2.0-sec thrust period followed. After the fifth sequence, a second burn was made at the high thrust levels used to assist in separating the retrorocket after burnout. Following another sequence of five 0.2-sec firings, a final attempt was made to open the fuel pressure regulator valve by commanding high thrust for 20 sec. The gas jets were enabled as part of this command sequence.

4. Termination of Mission

At approximately 09:30 on September 22, a terminal sequence was initiated. An emergency altitude-marking-radar mark was transmitted, and, after an 8-sec programmed delay, the vernier and retrorocket ignition programmer latches were set. At retrorocket ignition, the indicated acceleration level rose to 10.27 *g* and later rose to 11.72 *g*, at which time all data were lost.

C. Systems Engineering and Testing

1. SC-3 (Third Flight Spacecraft)

The third test sequence (plugs-out configuration) of mission sequence/electromagnetic interference (EMI) testing was designed to simulate the effects of the EMI environment to be encountered by the spacecraft before and during launch and from the *Atlas/Centaur* configuration. The sequence began with a performance verification test to verify proper operation between the systems test equipment assembly and the SC-3 spacecraft. The test data indicated that SC-3 was compatible and not susceptible to any EMI/RF interference irradiation. The spacecraft operated without any malfunction or degradation of performance during this test sequence.

The first phase of solar-thermal-vacuum testing began September 3. Test conditions were 87% solar intensity, chamber pressure of $<2.8 \times 10^{-6}$ torr, and chamber floor and wall temperatures of $< -300°F$. One major anomaly concerned phase jitter; another, which appeared to be a radar altimeter and doppler velocity sensor (RADVS) failure during terminal descent, was later found to be a problem in the systems test equipment assembly.

For the second phase of testing, the solar intensity was increased to 112%. No major spacecraft anomalies were encountered. However, during preparations for the third phase of solar-thermal-vacuum testing, an inadvertent turn-on of the RADVS necessitated replacement of the klystron power supply modulator and signal data converter units. Transmitter A was also replaced due to the phase jitter problem encountered in the first test phase. The vernier propulsion system was removed for rework and replacement of several units and components.

Due to these replacements, operational plans for the third phase of testing were modified. The flight-control closed-loop hardware was reinstalled so the new RADVS units would be thoroughly tested in the solar-thermal-vacuum environment. The duration of the test was extended to 50 hr for proper environmental exposure of the RADVS, the new transmitter, and particularly the vernier propulsion system. Solar intensity levels were altered to provide a low-/high-intensity test with approximately an equal amount of exposure at each level.

The third phase of solar-thermal-vacuum testing (plugs-out configuration) began October 6. Due to a cutoff of the Transmitter B voltage-controlled crystal oscillator during transfer from high- to low-power modes and below-minimum temperatures in one oxidizer tank, a retest was scheduled.

The next tests scheduled for SC-3 are the spacecraft and vernier engine vibration tests. Test configurations will differ from those used during vibration tests of the SC-2 (*Surveyor II*) spacecraft. Preparations for the vibration tests began in parallel with the solar-thermal-vacuum test phases. Thermal-simulation dummies of the soil mechanics/surface sampler mirrors and their substructure (discussed in Section G) were installed in time for the third phase of solar-thermal-vacuum testing (Fig. 1). Fabrication is under way on dynamically similar dummy units with flight-quality substructure to be installed in time for a portion of SC-3 vibration testing.

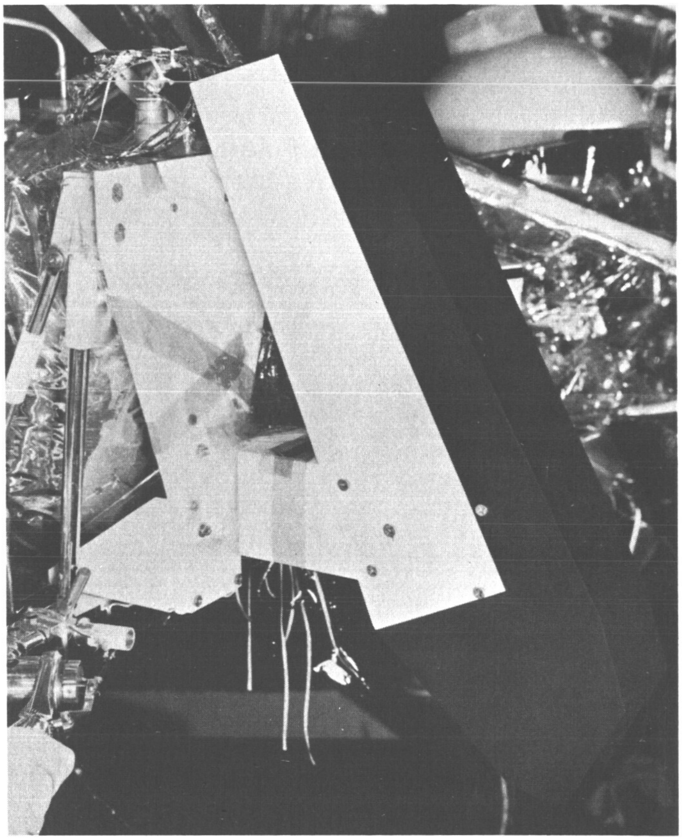

Fig. 1. Soil mechanics/surface sampler mirrors and substructure installed on SC-3

2. SC-4 (Fourth Flight Spacecraft)

The initial systems checkout test phase of SC-4, partially completed during the preceding reporting period, was completely rerun to meet the requirements of the revised systems test specification. Three SC-4 transmitters were removed to support SC-3 tests, causing a rerun of telecommunications integration efforts. After a mandatory RADVS rework at the vendor's facility, the mission sequence/EMI test phase was initiated on October 5. By the end of this reporting period, SC-4 was being prepared for solar-thermal-vacuum tests. A soil mechanics/surface sampler and mirrors of the same type as those used on SC-3 will also be installed on this vehicle.

3. SC-5 Through -7 (Fifth Through Seventh Flight Spacecraft)

The addition of the alpha-scattering instrument on the SC-5, -6, and -7 spacecraft necessitated modification of the basic bus to provide compatible interfaces between the spacecraft and the instrument. These modifications provide power switching, command decoding, signal processing, clock signals, and instrument uncaging and deployment. To provide the necessary interface, an integrated approach was selected. To implement this approach, it was necessary to remove the auxiliary battery and its compartment, substructure, harness, and the auxiliary control unit. The w-hr capacity at touchdown and the terminal-descent voltage of the main battery alone appear to be adequate. Additional analysis regarding the terminal-descent voltage of the main battery is required, however, to assured a satisfactory level. Also, battery tests at the new expected mission conditions (temperatures and loads) are essential.

The integrated approach requires no new hardware design, but only basic bus harness changes. No additional circuitry for mechanizing the commands or data channels is required either. Each of the present scientific auxiliaries contains a subsystem decoder that will be used to generate the required commands. The data channels required for the alpha-scattering instrument will be spare channels and those made available by the deletion of the approach TV camera and the auxiliary battery. Another set of data channels will also be used. In addition to supplying the required channels for the alpha-scattering instrument, these channels will optimize the present channel assignments by reducing the number of commutator mode changes during the missions and providing more meaningful data during the critical phases of the missions.

New substructure is being installed in the area of the auxiliary battery to support the alpha-scattering instrument sensor head and deployment mechanism. The instrument's electronics and auxiliary will be mounted in a new thermally controlled Compartment C above and slightly behind the nitrogen tank. A new payload harness will connect this compartment to Compartment B through a disconnect plug on the exterior adjacent to Compartment B.

In addition, mirrors similar to those on SC-3 and -4 will permit auxiliary viewing by survey TV Camera 3. One mirror will be available, if required, to view the lunar surface upon which the alpha-scattering instrument sensor head is deployed. The other mirror will be available to perform the same function as on SC-3 and -4, namely, to view the surface disturbed by the thrust chamber assembly or crushable blocks during landing.

The SC-5 spacecraft has been delivered for limited initial system checkout tests, after which it will be upgraded to accept the new payload for the remainder of the flight-acceptance testing. SC-6 is nearing the completion of substructure installation (Fig. 2), and substructure installation on SC-7 has begun.

D. Flight Control

Tests were performed on the roll actuator to investigate the effects of temperature on certain performance parameters, since an abnormally high starting potential for the actuator drive motor occurred during flight-acceptance testing of one of the roll actuator units. A minimum backlash at the second-stage gear mesh was established as the probable cause for the higher voltage required. These tests were felt desirable since no opportunity to determine temperature effects on a unit with minimum backlash existed previously. Flight-acceptance-test levels (minimum starting potential and the velocity test) were used, except that, in addition to the ambient run, runs were made at operating temperature limits of 0 and 200°F.

Test results showed that the minimum starting potential increased from 4.1 v at room temperature to 8.3 v at 0°F and dropped to 1.5 v at 200°F. Velocity showed no significant change and remained well within specification. Due to the effects of gear eccentricities, these results were as expected. However, no degradation in dynamic performance was indicated.

To further investigate temperature effects, a type-approval roll actuator unit was tested for stall torque over the same temperature range. The result that the stall torque measured highest at low temperature was considered normal because of the greater current flow through the motor at lower temperatures. The increase in torque over the temperature range was approximately 15%.

E. Power

During the *Surveyor I* coast phase, the auxiliary battery temperature reached a low of 35°F. In the event of a main battery failure, it is doubtful that the auxiliary

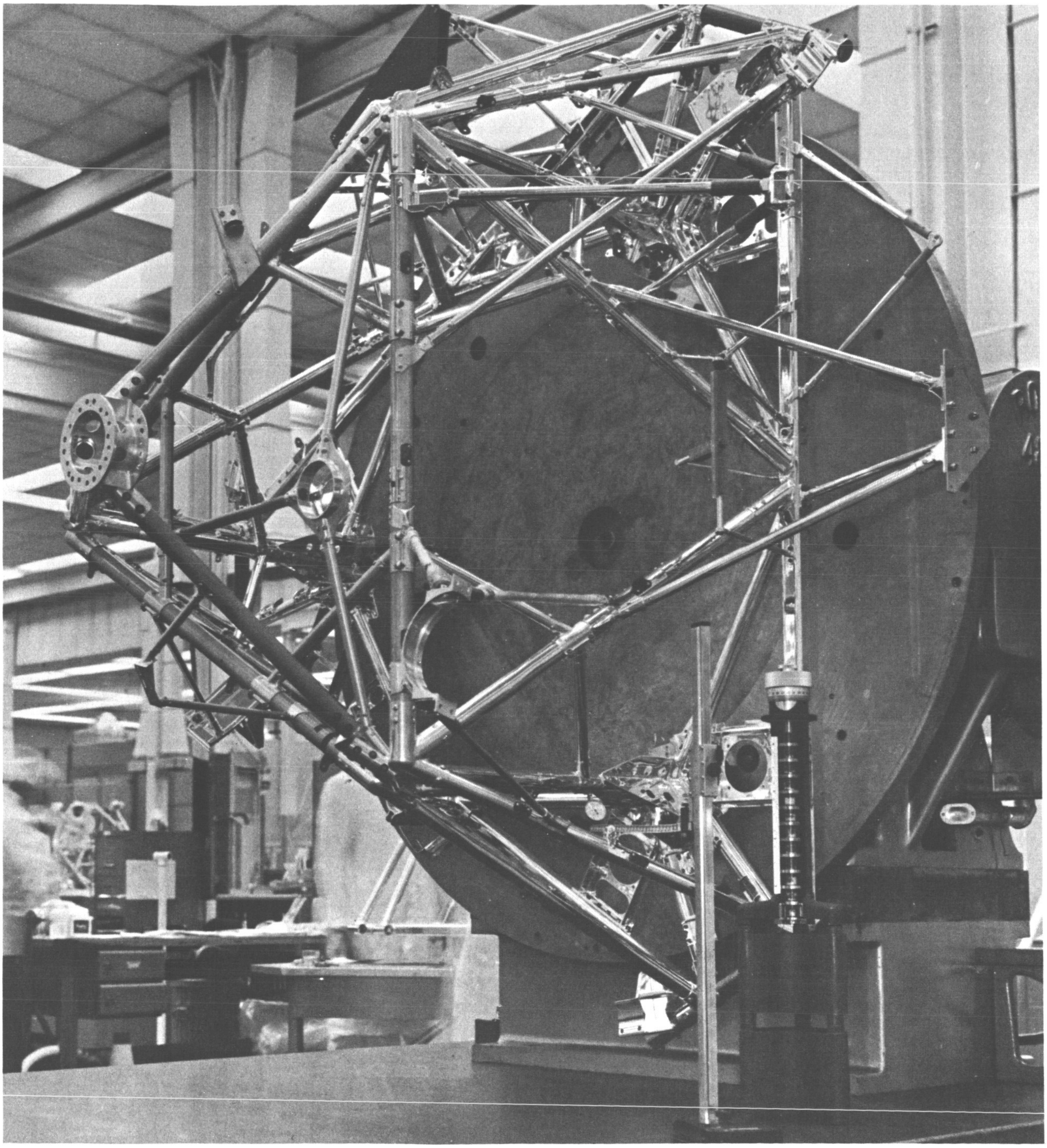

Fig. 2. SC-6 substructure installation

battery temperature would be high enough for the battery to maintain an adequate terminal voltage during terminal descent. Therefore, the thermal paint pattern of the auxiliary battery was changed to provide operating temperatures within the range of 40 to 130°F, preferably 60 ± 15°F during the coast phase and 95 ± 15°F at the beginning of terminal descent. The new paint pattern was tested on SC-3 during its solar-thermal-vacuum tests at both low and high solar intensity levels. Battery temperature during both tests was well within the predicted range.

The redesigned auxiliary battery cover was utilized on the *Surveyor II* spacecraft. The redesigned structure for the SC-6 auxiliary battery control has been completed. This new structure, which uses honeycomb instead of foam, is stronger and thinner than the previous structure.

F. Propulsion

The final reduction of *Surveyor I* flight data gave 38.9 ± 0.10 sec as the retrorocket engine burning time. This value differs from the nominal predicted value (38.5 sec) input to the spacecraft by slightly more than the ±1% allowed for retrorocket performance prediction inaccuracy. It is, however, within the over-all ±3% system tolerance provided. The *Surveyor I* vernier propulsion system was found to have performed satisfactorily in all respects.

Telemetry data from the *Surveyor II* premidcourse period indicates that the temperature gradient in the main retrorocket engine was developing in the same manner as that during the *Surveyor I* mission. Due to the tumbling action that followed the midcourse maneuver, retrorocket temperatures exceeded the predicted levels by approximately 16 to 25°F at the time of the commanded retrorocket ignition. Telemetry data received for the first 32 sec of retrorocket firing indicate that performance was what would be expected for the existing temperature conditions (thrust very near 10,000 lb).

A JPL–HAC investigation is under way to determine why vernier Engine 3 did not fire. Certain anomalies occurring in the telemetry data (e.g., a propellant shutoff valve command, propellant line and thrust chamber assembly temperature indications, and propellant flow) are being studied. Results will be presented when they are available.

G. Payload and Scientific Mechanisms

1. Flat Beryllium Mirrors

Information on the extent and nature of lunar surface disturbance and the amount of crushing of the spacecraft blocks after a *Surveyor* spacecraft landing could be used to determine lunar surface properties. To enable such a determination, two flat beryllium mirrors are being installed on SC-3 and -4. These mirrors will be attached to the inner portion of the spaceframe (Fig. 3) in such a

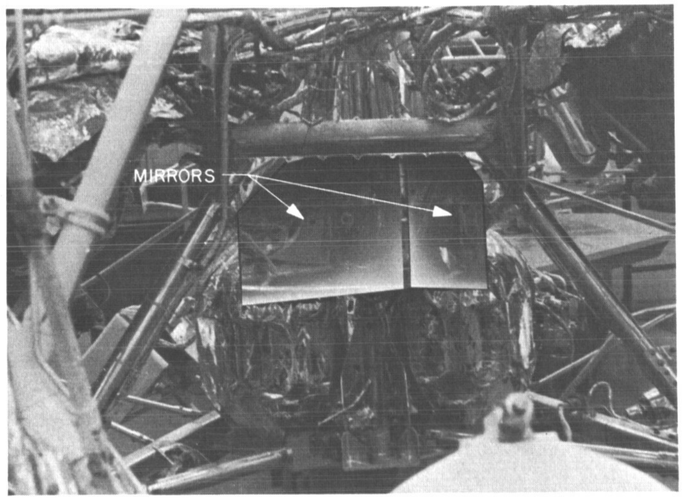

Fig. 3. Mirrors installed on inner portion of spaceframe

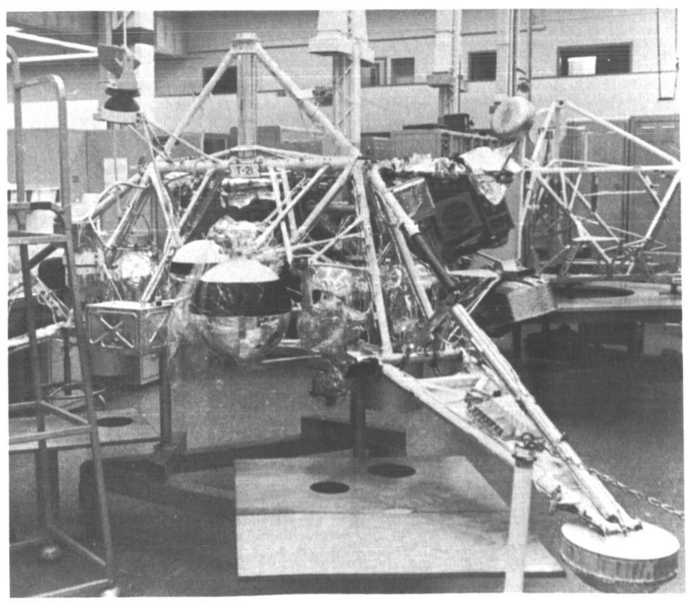

Fig. 4. Mirror viewing areas (dark circles)

manner that one will view a portion of the crushable blocks, the lunar area under both crushable blocks, and vernier Engine 3. The other mirror will view the lunar surface under vernier Engine 2. The dark circles shown below the spacecraft in Fig. 4 represent the areas to be viewed by the mirrors. By correctly positioning the survey TV camera in both elevation and azimuth, a TV picture may be taken of the two mirrors and, thus, of the lunar surface.

2. Soil Mechanics/Surface Sampler Instrument

A simplified soil mechanics/surface sampler instrument has been developed for incorporation on SC-3 and -4. With this instrument, a portion of the lunar surface near the spacecraft will be manipulated, and the results of the operations will be viewed by means of survey TV Camera 3. The visual data obtained during the picking, digging, scraping, and trenching operations, combined with a determination of the forces developed during trenching, will provide significant information relating to the characteristics of the lunar surface.

The instrument will be mounted to the spaceframe in place of the approach TV Camera 4 by using the existing substructure and adding a new mounting adapter. The electrical interface will be established at the Camera 4 interface connector, and all interconnections between the auxiliary and the spacecraft will be through the existing Camera 4 harness. Logic circuitry will be added to the soil mechanics/surface sampler auxiliary to receive the four Camera 4 command inputs and provide 16 command outputs to the instrument. A separate, newly designed, temperature-control compartment will house the instrument's auxiliary. This auxiliary will provide command decoding, buffering of instrument data, power management, squib firing, and total control electronics for the instrument. The auxiliary in its compartment will be located on the lower spaceframe member in spacecraft Sector 3.

The schedule objective is to design, develop, and flight-acceptance-test the SC-3 flight unit in time for delivery to the Eastern Test Range prior to SC-3 encapsulation for launch.

II. *Mariner* Venus 67 Project

THE PLANETARY–INTERPLANETARY PROGRAM

A. Introduction

The primary objective of the *Mariner* Venus 67 Project is to conduct a flyby mission to Venus in 1967 to obtain scientific information which will complement and extend the results obtained by *Mariner II* relevant to determining the origin and nature of Venus and its environment. Secondary objectives are to: (1) acquire engineering experience in the conversion of a spacecraft designed for a mission to Mars (spare flight spacecraft from *Mariner* Mars 1964 Project) into one designed for a mission to Venus and in the operation of such a spacecraft, and (2) obtain information on the interplanetary environment during a period of increasing solar activity. An *Atlas/Agena D* launch vehicle will be used.

Due to the minimum length of time (18 months) between authorization of the project and the launch opportunity, techniques and hardware developed during prior projects must be utilized to the fullest extent possible. The single flight spacecraft, designated M67-2, will be a converted *Mariner* Mars 1964 Project flight spare. Portions of the proof test model and certain critical spare units from the *Mariner* Mars 1964 Project are being prepared for use as a flight support spacecraft, designated M67-1. The flight support spacecraft will serve the double function of a pseudo-proof test model and a backup spacecraft for qualifying spare subsystems.

Various changes to the *Mariner* Mars 1964 spacecraft design are necessitated by the fact that the M67-2 flight spacecraft will travel toward, rather than away from, the Sun; also, conversions must be made to accommodate the revised encounter sequencing and science payload.

The S-band radio occultation experiment, one of seven scientific experiments approved for the mission, requires the use of only the RF transmission subsystem on the spacecraft. The celestial mechanics experiment uses only the tracking doppler data derived from the RF carrier. The ultraviolet photometer, helium magnetometer, solar plasma probe, and trapped radiation detector experiments are to be accomplished using existing instrumentation with only minor modifications. Only the dual-frequency radio propagation experiment requires the incorporation of a new scientific instrument into the payload. The Principal Scientific Investigators for these experiments are given in Table 1.

During this reporting period, delivery of the remaining flight-quality telecommunications and science hardware continued. Once this phase has been successfully completed, the spacecraft environmental test cycle will be initiated. The flight support spacecraft M67-1, with some prototype engineering units and without the science subsystem installed, completed its first systems test.

Table 1. *Mariner* Venus 67 Principal Scientific Investigators

Experiment	Principal Scientific Investigator	Affiliation
S-band radio occultation	A. J. Kliore	Jet Propulsion Laboratory
Ultraviolet photometer	C. A. Barth	University of Colorado
Dual-frequency radio propagation	V. R. Eshleman	Stanford University
Helium magnetometer	E. J. Smith	Jet Propulsion Laboratory
Solar plasma probe	H. S. Bridge	Massachusetts Institute of Technology
Trapped radiation detector	J. A. Van Allen	State University of Iowa
Celestial mechanics	J. D. Anderson	Jet Propulsion Laboratory

B. Design and Development

1. Systems-Test-Complex Data System

A computer-based data system is under development for use in acquiring, processing, recording, and displaying real-time data acquired during *Mariner* Venus 67 spacecraft systems tests. In early-October, the capability of the system to acquire, record, and process spacecraft telemetry data was demonstrated. This processing includes a capability to suppress, on any data channel, the printout of all data samples whose values have changed less than some predetermined amount from the previously displayed values. Other capabilities provided are limit checking, conversion to engineering units, and the changing of tolerances and limits used in processing by card control in real time. Several routines for the special processing of telemetry were also demonstrated: calculation of system and subsystem power, command detector status, temperature measurement corrections, and the decommutation and formatted display of science telemetry (excluding encounter telemetry, which is reproduced from the tape recorder in the encounter data mode).

Installation of the computer system and the data display subsystem was completed, and additional elements of the data input subsystem are being installed and checked out. The analog modules for measurement of spacecraft currents and voltages are in the early design phase. Additional functional requirements which have been established (the real-time acquisition, processing, recording, and display of the science data automation subsystem outputs to the data encoder and tape recorder subsystems), together with the requirement for processing two spacecraft telemetry data streams simultaneously, necessitated the procurement of four additional data input modules for the data input system.

2. Central Computer and Sequencer (CC&S)

The *Mariner* Mars 1964 Project's engineering-prototype and type-approval CC&S subsystems have been modified for use on the *Mariner* Venus 67 spacecraft. After environmental tests of the modified unit for the M67-1 spacecraft were completed, the unit was delivered to the JPL Spacecraft Assembly Facility. The unit for the M67-2 flight spacecraft underwent environmental tests; the time-base oscillator was calibrated; and this unit was also delivered to the Spacecraft Assembly Facility.

Work was begun on the digital-to-analog converter for flight analysis support. A major redesign of the *Mariner* Mars 1964 Project's digital-to-analog equipment was necessitated by a change in the JPL Space Flight Operations Facility data format (both high speed and teletype). Fabrication of the first unit was initiated during this reporting period.

3. Attitude-Control Subsystem

A minimum number of changes were made to the *Mariner* Mars 1964 attitude-control subsystem to adapt it for use on the *Mariner* Venus 67 spacecraft. Among the required changes were the following:

(1) Due to the higher temperatures expected on the Venus mission, the gas system was reworked and is now qualified to operate in the temperature range of -45 to 215°F. As part of the rework, the solar vanes and associated cabling were removed.

(2) The Earth detector was relocated on the bus, and the baffling now restricts the field of view so that the Earth may be used as a crude roll reference (in the event of failure of the Canopus sensor) for up to 20 days after launch. Since the Earth detector has now been assigned its own telemetry channel, telemetered indications of the Earth's position in the field of view may be used to command spacecraft turns (± 2.27 deg) to fix the spacecraft's position prior to the midcourse maneuver.

(3) The location and cone angles of the Canopus sensor were redefined. The Sun shutter was moved inside the baffling, and the baffling was redesigned to preclude a Venus stray-light problem. Some suspect

parts were removed from the electronics, and the $\times 8$ Canopus intensity gate was removed to prevent transient loss of Canopus track due to small particles entering the field of view. The Sun shutter solenoid was redesigned to reduce magnetic interference.

(4) The narrow-angle Mars gate electronics circuitry was modified to provide an output to the pyrotechnics subsystem when the terminator of Venus enters the sensor's field of view. This output will trigger the mechanism which changes the high-gain antenna pointing angle.

(5) The primary Sun sensor assemblies were relocated on the sunlit side of the bus, and the secondary assemblies were moved to the shadowed side.

(6) A new sensor, the planet sensor, was incorporated. This device will initiate the scientific experiments upon sensing the limb of Venus (approximately 1 hr before encounter).

4. Thrust Vector Control Assembly

The thrust vector control assembly for the *Mariner* Venus 67 spacecraft consists of four jet vane actuators mounted on a ring, as shown in Fig. 1. The ring holds the vanes in the exit plane of the thrust vector control motor. Autopilot control forces for the spacecraft are provided in all three axes (pitch, yaw, and roll) by the vanes deflecting the exhaust gasses from the motor during the spacecraft midcourse maneuver.

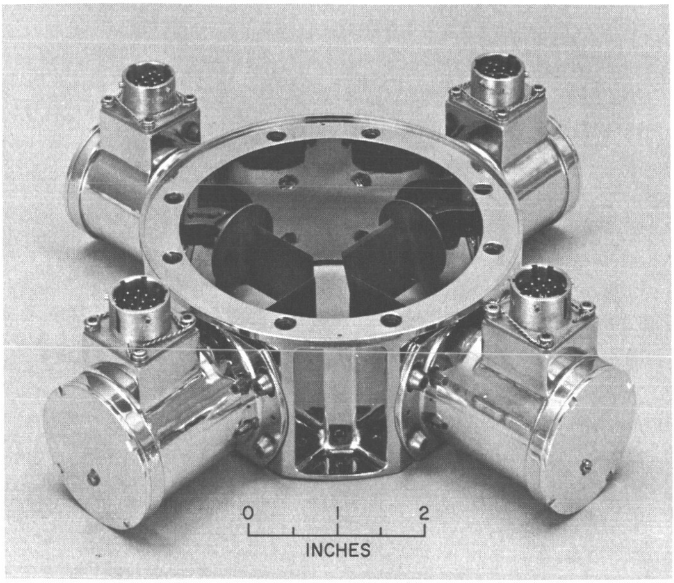

Fig. 1. Thrust vector control assembly

Eight jet vane actuators (two rings) manufactured and tested for use on the *Mariner* Mars 1964 spacecraft were re-evaluated for use on the *Mariner* Venus 67 spacecraft. All failed initial bench tests because of high stiction. After rework, the actuators underwent flight-acceptance testing, and all failed to operate at 32°F due to shaft binding. The actuators were reworked a second time, and the type of shaft seal lubricant was changed. The second attempt at flight-acceptance-testing these actuators was successful. One flight-qualified thrust vector control assembly has been delivered to the Spacecraft Assembly Facility.

III. *Mariner* Mars 1969 Project

THE PLANETARY–INTERPLANETARY PROGRAM

A. Introduction

The primary objective of the *Mariner* Mars 1969 Project is to conduct two flyby missions to Mars in 1969 to make exploratory investigations of the planet which will set the basis for future experiments—particularly those relevant to the search for extraterrestrial life. The secondary objective is to develop the technology needed for succeeding Mars missions.

The spacecraft design concept will be based on that of the successful *Mariner IV* spacecraft developed under the *Mariner* Mars 1964 Project. However, considerable modifications will be made to meet the 1969 mission requirements and to enhance mission reliability.

The launch vehicle will be the *Atlas/Centaur SLV-3C*. This vehicle, developed under contract for and direction by the Lewis Research Center by General Dynamics/Convair, has a single- or double-burn capability in its second stage and a considerably increased performance rating over the *Atlas D/Agena D* used in the *Mariner IV* mission.

Mariner Mars 1969 missions will be supported by the Eastern Test Range launch facilities at Cape Kennedy, the tracking and data acquisition facilities of the Deep Space Network, and other NASA facilities.

The six planetary-science experiments tentatively selected by NASA (subject to integration capability) for the *Mariner* Mars 1969 missions are the following: TV, infrared spectrometer, ultraviolet airglow spectrometer, infrared radiometer, S-band occultation, and celestial mechanics. Additionally, a planetary-approach-guidance engineering experiment will be incorporated to test the feasibility and flightreadiness of onboard optical sensors and associated data processing techniques necessary for optical approach guidance. The Scientific Investigators for the planetary-science experiments are listed in Table 1.

During this reporting period, selection of subsystem contractors and mission and spacecraft system design efforts continued. The exploration of the scientific instrument requirements and their interaction affected the design efforts.

Table 1. *Mariner* Mars 1969 Scientific Investigators

Experiment	Scientific Investigator	Affiliation
TV	R. B. Leighton[a]	California Institute of Technology
	B. C. Murray	California Institute of Technology
	R. P. Sharp	California Institute of Technology
	N. H. Horowitz	California Institute of Technology
	J. D. Allen	Jet Propulsion Laboratory
	A. G. Herriman	Jet Propulsion Laboratory
	L. R. Malling	Jet Propulsion Laboratory
	R. K. Sloan	Jet Propulsion Laboratory
	N. Davies	Rand Corporation
	C. Leovy	Rand Corporation
Infrared spectrometer	G. C. Pimentel[a]	University of California, Berkeley
	K. C. Herr	University of California, Berkeley
Ultraviolet airglow spectrometer	C. A. Barth[a]	University of Colorado
	W. G. Fastie	Johns Hopkins University
Infrared radiometer	G. Neugebauer[a]	California Institute of Technology
	G. Munch	California Institute of Technology
	S. C. Chase	Santa Barbara Research Center
S-band occultation	A. J. Kliore[a]	Jet Propulsion Laboratory
	D. L. Cain	Jet Propulsion Laboratory
	G. S. Levy	Jet Propulsion Laboratory
Celestial mechanics	J. D. Anderson[a]	Jet Propulsion Laboratory

[a] Principal Scientific Investigator.

B. Mission Profile

The *Mariner* Mars 1969 mission profile may be divided into the following five phases:

1. Launch Phase

The spacecraft will be boosted into an interplanetary transfer trajectory by its *Atlas/Centaur SLV-3C* launch vehicle. The launch profile will be direct ascent. Since two traveling-wave-tube power amplifiers will be incorporated in the radio subsystem (for which no low-power mode is included), the present mission profile assumes that the traveling-wave-tube power will be on and radiating 10 w of RF power from liftoff.

The spacecraft will be separated from the *Centaur* on command from that stage. Following separation, the spacecraft will acquire the Sun and immediately commence Canopus acquisition. No magnetometer roll calibration such as that performed during the *Mariner IV* mission will be required, since no magnetometer is included in the *Mariner* Mars 1969 science payload. An adaptive gate for star brightness in the Canopus sensor will be successively lowered by cyclic command from the central computer and sequencer (CC&S) until a celestial object, hopefully Canopus, is acquired.

2. Cruise Phase

Since the science payload contains no instruments to be operated during the cruise phase, the data mode for the launch and cruise phases (called the cruise data mode) consists solely of engineering data. At the beginning of cruise, the spacecraft will transmit its engineering data at 33⅓ bits/sec on its low-gain antenna. The cone angle of the Canopus sensor will be updated by programmed command by the CC&S during the flight. When communications performance drops to the sum of the negative tolerances, the CC&S will command the telemetry subsystem to lower the bit rate to 8⅓ bits/sec. Several days after the communication performance again drops to the sum of the negative tolerances, the CC&S will command the radio subsystem to switch the power amplifier to the fixed high-gain antenna. At the 8⅓-bit/sec rate when transmitting on the high-gain antenna, the communications performance will improve rapidly. After a suitable performance margin has been obtained, the telemetry bit rate will be switched to 66⅔ bits/sec, the rate for the encounter mode and first part of the playback mode.

The approach-guidance subsystem will be turned on 10 days prior to arrival at Mars, this being the last event of the cruise phase. From that time until 12 hr before encounter, the center of Mars will be optically tracked, with the cone and clock angles measured in the spacecraft reference system. These angles and the Sun and Canopus sensor error signals will be telemetered to Earth for processing and subsequent use in the orbit-determination program.

Early in the flight, the cruise phase will be interrupted to perform a trajectory correction maneuver. As in the *Mariner* Mars 1964 design, the *Mariner* Mars 1969 design provides a capability for a second maneuver, should such a maneuver be necessary.

3. Maneuver Phase

In the tandem maneuver mode, the CC&S will compare the output of the programmable sequencer against the output of the fixed sequencer. If a discrepancy exists, the maneuver will be automatically aborted. The capability will exist to select, by ground command, the correctly functioning sequencer and to repeat the maneuver sequence.

4. Encounter Phase

The encounter phase of the *Mariner* Mars 1969 missions will be divided into three sequences: far-encounter, near-encounter, and occultation. At the beginning of the far-encounter sequence, the scan control, TV, infrared radiometer, and data automation subsystems will be energized. After the science instruments have warmed up and sufficient time has elapsed for the evaluation of spacecraft performance in the far-encounter mode, the scan platform will be commanded from the erected near-encounter position (cone angle ≅ 120 deg) to the far-encounter position. If the planet brightness is above the threshold for the far-encounter scan sensor, the scan control subsystem will switch from open-loop pointing to a closed-loop mode and servo on the planet. A series of TV pictures will be taken by the narrow-angle TV camera from 48 hr to 12 hr before closest approach to Mars. The picture data will be stored on a *Mariner* Mars 1964-type tape recorder, using an analog recording scheme. At the conclusion of the far-encounter sequence, the scan platform will be commanded back to the near-encounter position.

At the beginning of the near-encounter sequence several hours prior to closest approach to Mars, all science instruments will be energized. The scan platform will be fixed at a 120-deg cone angle and at one of the selectable clock positions. (The clock position can be stepped by ground command through several discrete positions in 3-deg increments.) Thus, no servo sensor or closed-loop planet tracking is presently planned for the near-encounter sequence.

At 20 min prior to the start of data taking, a CC&S command will initiate cooling of the infrared spectrometer detectors. When the planet comes within view of the narrow-angle TV camera, near-encounter data taking will commence. During the data-taking period, the scan platform cone angle will be dropped from 120 to approximately 90 deg to increase the duration of data taking. The data-taking period will continue until both the analog and digital recorders are full.

Both the near- and far-encounter sequences will be conducted in the encounter data mode at 66⅔ bits/sec. This data mode is analogous to the *Mariner* Mars 1964 cruise data mode in that 140 bits of engineering data will be followed by 280 bits of science data. No data mode equivalent to the *Mariner* Mars 1964 all-science data mode will be provided.

Approximately 1 min after data taking has ceased, the spacecraft will enter the occultation region at Mars and will be occulted from Earth for approximately 30 min. After the spacecraft's exit from the region, several hours will be required for transmission of engineering data to enable the analysis of spacecraft performance prior to the start of the playback phase.

Table 2. Comparison of *Mariner* Mars 1964 and *Mariner* Mars 1969 spacecraft weights

Item	Weight, lb	
	Mariner Mars 1964	*Mariner* Mars 1969
Science		
Platform	9.5	57.0
Bus	42.4	69.0
Data storage	17.1	36.0
Engineering	506.6	574.5
Spacecraft adapter	—	50.0
Contingency	—	57.6
Total	575.6	844.1

5. Playback Phase

The bit rate for the playback phase of the mission will be 66⅔ bits/sec until the communications performance starts to degrade, at which time the bit rate will be commanded to 33⅓ bits/sec. The latter rate will be in effect for the remainder of the mission. The present mission plan calls for one playback of the recorded data. Assuming 2 hr of engineering telemetry per day and at the 66⅔-bit/sec rate, 35 days will be required for playback. This phase of the *Mariner* Mars 1969 missions will be more complex than the playback phase of the *Mariner IV* mission in that periodically (approximately every 3 days when transmitting at 66⅔ bits/sec) data will be transferred from the analog tape recorder, through an analog-to-digital converter, to the digital tape recorder.

C. Design and Development

1. Spacecraft Subsystems

The technical approach necessary to satisfy the *Mariner* Mars 1969 mission objectives necessitates major changes to the *Mariner* Mars 1964 spacecraft subsystems. A summary of the required design changes is presented here.

a. Structure. The basic structure must be modified to accommodate the changes to other subsystems and the over-all increase in spacecraft weight from the *Mariner* Mars 1964 configuration. A weight comparison is given in Table 2. Also, a new adapter section for mating the spacecraft to the launch vehicle is required to accommodate the change of the launch vehicle second stage from an *Agena* to a *Centaur*. Included in this change is the relocation of the umbilical connector. The adapter uses the *Mariner* Mars 1964 design at the separation plane and the *Surveyor* design at the field joint. The use of a *Surveyor*-type shroud requires that the spacecraft umbilical lines be routed down through the adapter and out the *Centaur* umbilical island.

b. Mechanical devices. The mechanical devices subsystem is newly designated, incorporating several functions previously assigned to other subsystems. Specifically, this subsystem will include the solar panel boost damper; low-gain antenna damper; solar panel deployment and latch mechanisms, including the switch assemblies for indications of the deployment-initiated timer; spacecraft separation mechanisms; spacecraft V-band; and the scan platform. Nearly all of these devices require modification for *Mariner* Mars 1969 spacecraft use. The scan platform must be redesigned to incorporate 2 deg of freedom and accommodate a six-fold increase in the weight of the instruments mounted on it. The two-camera TV, ultraviolet and infrared spectrometers, and infrared radiometer will be supported by the scan platform.

c. Temperature control. The temperature-control subsystem will probably require heater power for the scan platform. A temperature-control flux monitor has been incorporated in lieu of the *Mariner* Mars 1964 absorptivity standard.

d. Pyrotechnics. Certain modifications to the pyrotechnics subsystem will be necessary to meet new requirements imposed by other subsystems. For example, release devices required for spacecraft separation and squib-actuated valves for activation of the infrared spectrometer cool-down gas supply will be added. In the pinpullers, 1-amp/1-w no-fire squibs will be utilized. The number of pinpullers, unlatching events, and the number of pinpullers per event will be revised to meet the requirements of system design.

e. Propulsion. At present, no changes to the *Mariner* Mars 1964 propulsion subsystem design are planned.

f. Power. The complete power subsystem, including boost regulators, inverters, battery charger, and power distribution logic, will be redesigned. The power subsystem shown in Fig. 1 has been the subject of design studies at JPL. A solar panel configuration analysis has indicated that four rectangular panels, each measuring 35.5×84.2 in. for a combined active area of 83 ft^2, will fit within the *Surveyor*-type shroud. To provide increased protection against solar panel damage, N/P solar cells will be used. The new solar panels will not have solar vanes and will be tip-latched in the stowed position. The primary timing reference has been moved from the CC&S into the power subsystem. Structural changes require the incorporation of additional Sun sensors. The energy storage system will be an 18-cell, sealed, silver-zinc, alkaline battery weighing 34 lb.

g. Science. The mission objectives necessitate a completely new science subsystem which will include TV, a data automation system, an infrared spectrometer, an ultraviolet airglow spectrometer, and an infrared radiometer. The approach-guidance function requires a completely new subsystem. Since no magnetometers or particle detectors are included in the science subsystem, there is no requirement in the spacecraft design for magnetic cleanliness and absence of radioactive materials.

h. Radio. The radio subsystem will incorporate two traveling-wave-tube power amplifiers in place of the one traveling-wave-tube power amplifier and one cavity amplifier used in the *Mariner* Mars 1964 design. The receiver equipment has been moved from Bay V to Bay VI to make space available for two tape recorders in Bay V. To simplify the switching and eliminate any uplink antenna interference, the receiver will be attached to only the low-gain antenna.

i. Telemetry. Three data transmission rates will be available: 8⅓, 33⅓, and 66⅔ bits/sec. The latter bit rate will be possible since the communications distance for the *Mariner* Mars 1969 spacecraft encounter will be smaller than that for the *Mariner* Mars 1964 spacecraft encounter. Also, new data modes have been incorporated. The telemetry channel assignments have been changed to accommodate the new requirements of the other subsystems. A requirement exists for the processing of digital data from approach guidance and the CC&S subsystem.

j. Command. In the Mariner Mars 1969 command subsystem, the isolated switch outputs have been reallocated among the direct commands (DCs).

k. Data storage. Due to the large increase in the required amount of data storage (caused by not only the increased number of TV pictures, but also the increased number of lines and elements per picture), a new mechanization is proposed for the data storage subsystem. This

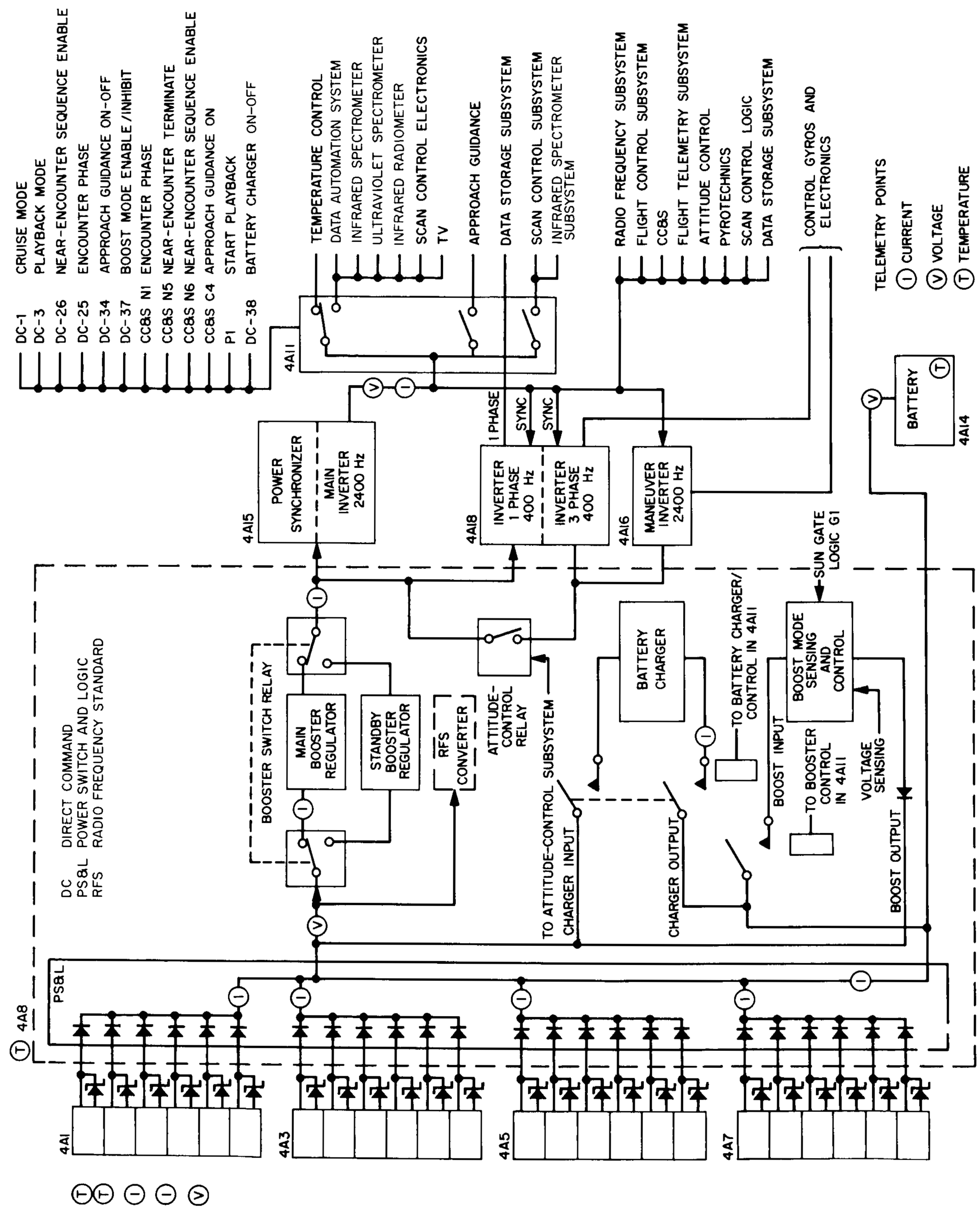

Fig. 1. *Mariner Mars 1969* **power subsystem**

new mechanization will permit a >30-fold increase in data storage capability: from 5.2×10^6 bits for the *Mariner* Mars 1964 mechanization to 1.8×10^8 bits for the *Mariner* Mars 1969 mechanization. The new mechanization consists of two *Mariner* Mars 1964-type endless-loop tape recorders, one to store 2.3×10^7 bits in digital form and the other to store 1.57×10^8 bits in analog form. Both the record and playback electronics have been changed, and an analog-to-digital converter has been added to digitize the analog data transferred to the digital tape recorder during the playback phase.

l. Attitude control. In the attitude-control subsystem, the Canopus sensor incorporates new star tracking logic, including an adaptive gate. The Earth detector has been deleted, and all attitude-control electronics have been repackaged.

m. Scan control. The scan-control subsystem is entirely new. Control is required for the 2-deg-of-freedom scan platform. For the far-encounter sequence, a closed-loop servo will be required; for the near-encounter sequence, only open-loop pointing will be used. The scan-control subsystem will also provide open-loop pointing for the ultraviolet airglow spectrometer.

n. CC&S. The *Mariner* Mars 1969 CC&S subsystem, a new design, is a programmable sequencer with a 128-word core memory that can be updated or modified by ground command. This sequencer will allow a versatility in mission operations heretofore unobtainable with previous designs. Redundancy will exist in the maneuver logic, since the CC&S will have a separate fixed sequencer for the maneuver functions.

2. Operational Support Equipment

The operational support equipment (OSE) discussed here is the ground support equipment that will provide the power, control, stimulation, monitoring, and interface simulation necessary for testing the spacecraft attitude-control, scan-control, approach-guidance, CC&S, and power subsystems. The first three of these subsystems will be supported by a single OSE unit called the spacecraft control OSE. The CC&S and power subsystems will be supported by separate units.

Because the *Mariner* Mars 1969 spacecraft is an adaptation and expansion of the *Mariner* Mars 1964 and *Mariner* Venus 67 spacecraft configurations, much of the OSE used for testing these spacecraft will be modified for use on the *Mariner* Mars 1969 Project. The power OSE requires only modification to the existing test sets. The spacecraft control OSE requires new assemblies for scan and approach-guidance support, as well as modifications to the existing attitude-control OSE. New OSE is being designed and built to support the new computer-oriented CC&S configuration of the *Mariner* Mars 1969 spacecraft.

IV. *Voyager* Project

THE PLANETARY–INTERPLANETARY PROGRAM

A. Introduction

1. Objectives

The primary objective of the *Voyager* Project is to carry out scientific investigations of the solar system by instrumented, unmanned spacecraft which will fly by, orbit, and/or land on the planets. Emphasis will be placed on the acquisition of scientific information relevant to the origin and evolution of the solar system and the origin, evolution, and nature of life; and the application of this information to an understanding of terrestrial life. The primary objective of the *Voyager* missions to Mars beginning in 1973 is to obtain information relative to the existence and nature of extraterrestrial life; the atmospheric, surface, and body characteristics of Mars; and the planetary environment by performing automated experiments on the surface of, and in orbit about, the planet. A secondary objective is to further our knowledge of the interplanetary medium between the planets Earth and Mars by obtaining scientific and engineering measurements while the spacecraft is in transit.

2. Project Plan

All *Voyager* missions will be conducted as events of an integrated program in which each individual flight forms a part of a logical sequence in an over-all technical plan of both lander and orbital operations. The *Voyager* design will provide for the transport of large scientific payloads to the planet, the telemetering of a large volume of data back to Earth, and long useful lifetimes in orbit about the planet and/or on the planetary surface. Hardware will be designed to accommodate a variety of spacecraft and/or capsule science payloads, mission profiles, and trajectories. Particular emphasis will be given to simple and conservative designs; redundancy wherever appropriate; and a comprehensive program of component, subsystem, and system testing.

Over-all direction and evaluation of the *Voyager* Project is the responsibility of the NASA Office of Space Science and Applications (OSSA). The project's management and the implementation of selected systems are the responsibilities of JPL.

Since no deep space flight tests of the flight spacecraft are planned, compensation will be made by more extensive ground tests conducted to the extent possible. During the test program for the flight capsule, which is planned to include Earth-entry flight tests, entry dynamics will be investigated.

3. Technical Description

Planning shall be based upon launching two identical planetary vehicles by a single *Saturn V* launch vehicle at each Martian opportunity, starting with the 1973 opportunity. Each planetary vehicle is to consist of a flight spacecraft, flight capsule bus, and surface laboratory, with science experiments conducted in 1973 from the orbiter, during descent, and on the planetary surface. The flight spacecraft with its capability for a 400-lb science subsystem (250 lb of instruments) will weigh approximately 2,500 lb; its retropropulsion subsystem may additionally weigh up to 15,000 lb; and the total flight capsule (capsule bus and surface laboratory) will weigh up to 5,000 lb. The flight spacecraft will be a fully attitude-stabilized device utilizing celestial references for the cruise phase. Velocity increments for midcourse trajectory corrections and for Mars orbit attainment by both the flight spacecraft and the flight capsule will be provided. Onboard sequencing and logic and ground command capability will also be provided. The flight spacecraft will supply its own power from solar energy or from internal sources and will be capable of maintaining radio communications with Earth. In addition, the flight spacecraft will be thermally integrated and stabilized. Data concerning various scientific phenomena near Mars and during transit and those concerning spacecraft performance will be monitored and telemetered to Earth.

The flight spacecraft will also provide the flight capsule with services such as power, timing and sequencing, telemetry, and command during the transit portion of the missions and may also serve as a communications relay. The sterilized flight capsule bus will be designed for separation from the flight spacecraft in orbit, attainment of a Mars impact trajectory, entry into the Martian atmosphere, descent to the surface, and soft landing the surface laboratory, which will have a surface lifetime of up to 2 days in 1973. Later goals for surface lifetimes are as much as 6 mo. The flight capsule bus will contain the propulsion, power, guidance, control, communications, and data handling systems necessary to complete its mission, while the surface laboratory will contain all power, sequencing, and communications for the conduct of surface operations.

B. Design and Development

1. Data Compression Evaluation Console

An experimentation program has been initiated in which an engineering data compression evaluation console (DCEC) will be used to test the performance of various system algorithms when used in conjunction with a zero-order predictor for data redundancy reduction. The DCEC, shown in Fig. 1, is organized as a piece of laboratory test equipment. The input medium will normally be digital magnetic tape read in by an integral tape system. Complete flexibility is available in the selection of

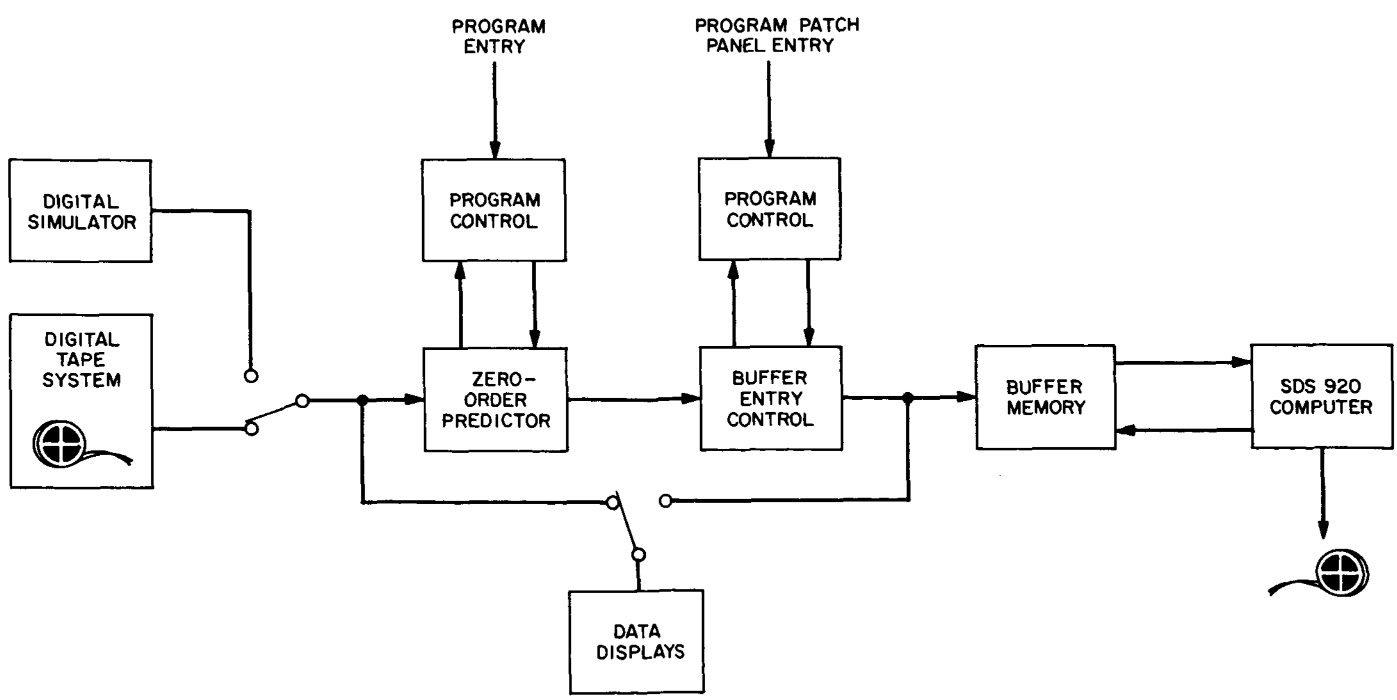

Fig. 1. DCEC block diagram

input data, since each file on a tape can be either a hybrid or a strictly digital computer simulation of peculiar or complex mode event periods of an unmanned spacecraft mission. Fourteen data channels can be simulated for each run. Provision is also made for a digital simulator to be used as an input medium for certain experiments calling for all channels to process the same waveform.

The input data will be fed to a zero-order predictor, where significance of each data point will be determined by employing a programmed control parameter. This parameter is an aperture code, separately programmed for each channel, which establishes how much a given sample value must deviate in amplitude from the last significant sample from that channel in order to be considered significant.

If a sample is considered significant, it is sent to the buffer memory input control circuitry, where the volume of data in the system buffer memory is monitored. The sample will be loaded into the buffer only if the criterion for the buffer entry control algorithm is satisfied. System output data will be read out upon command into a Scientific Data Systems (SDS) 920 computer, which will act as an output formatting and recording device. The data transfer rate will be programmed to simulate the relative rate of a spacecraft communications link.

Experiments with the DCEC should provide important information on: (1) the effects of data compression on various types and combinations of actual spacecraft system data waveforms, and (2) the extent to which the retrieval of necessary information is impaired by buffer limitations or queuing routines.

2. Spacecraft 50-w TWT Amplifier

The development of a spacecraft 50-w traveling-wave-tube (TWT) amplifier was reported in SPS 37-41, Vol. VI, pp. 22, 23. The original design called for a TWT with 22% beam power/RF power conversion efficiency (basic efficiency) and a collector operated at less than 50% of the helix voltage (depressed collector). Difficulty was experienced in obtaining proper operation with a depressed collector. Even though a basic efficiency of 25.6% was achieved, the over-all efficiency was only 31.3%. Also, saturated gain in the four TWTs fabricated has been low. Further efforts are being made to increase the basic efficiency, reduce the helix interception current (which limited the voltage depression attainable), and increase the gain.

The design of the power converter was essentially completed in early-October. Efficiency was within specification limits at -10 and 25°C. However, for some ripple frequencies, ripple sensitivity was calculated to be out of specification.

V. DSN Capabilities and Facilities

THE DEEP SPACE NETWORK

Established by the NASA Office of Tracking and Data Acquisition and under the system management and technical direction of JPL, the Deep Space Network (DSN) is responsible for two-way communications with unmanned spacecraft travelling from approximately 10,000 miles from the Earth to interplanetary distances. [Earth-orbiting scientific and communications satellites and the manned spacecraft of the *Gemini* and *Apollo* Projects are tracked by the Space Tracking and Data Acquisition Network (STADAN) and the Manned Space Flight Network (MSFN), respectively.] NASA space exploration projects supported, or to be supported, by the DSN include the following: (1) *Ranger, Surveyor, Mariner* Mars 1964, *Mariner IV, Mariner* Venus 67, *Mariner* Mars 1969, and *Voyager* Projects of JPL; (2) *Lunar Orbiter* Project of the Langley Research Center; (3) *Pioneer* Project of the Ames Research Center, and (4) *Apollo* Project of the Manned Spacecraft Center (as backup to MSFN).

Present DSN facilities permit simultaneous control of a newly launched spacecraft and a second spacecraft already in flight. In preparation for increased U.S. activities in space, a capability is being developed for simultaneous control of either two newly launched spacecraft plus two in flight, or four spacecraft in flight. Advanced communications techniques are being implemented to enable obtaining data from, and tracking spacecraft to, planets as far out in space as Jupiter. The main elements of the DSN are described below.

A. Deep Space Instrumentation Facility

The Deep Space Instrumentation Facility (DSIF) is composed of tracking and data acquisition stations around the world. The deep space stations (DSS's) of the DSIF and the deep space communication complexes (DSCC's) they comprise are as follows:

DSS	DSCC
Pioneer Echo Venus Mars	Goldstone
Woomera Tidbinbilla Booroomba[b]	Canberra
Johannesburg	
Robledo Cebreros[c] Rio Cofio[b]	Madrid[a]
Cape Kennedy (spacecraft monitoring)	
Ascension Island (spacecraft guidance and command)	

[a]Planned.
[b]Station not yet authorized.
[c]Station not yet operational.

These stations are situated such that three may be selected approximately 120 deg apart in longitude in order that a spacecraft in or near the ecliptic plane is always within the field of view of at least one of the selected ground antennas. JPL operates the U.S. stations and the Ascension Island DSS. The overseas stations are normally staffed and operated by government agencies of the respective countries, with the assistance of U.S. support personnel.

The Cape Kennedy DSS supports spacecraft final checkout prior to launch, verifies compatibility between the DSN and the flight spacecraft, measures spacecraft frequencies during countdown, and provides telemetry reception from liftoff to local horizon. The other stations obtain angular position, velocity (doppler), and distance (range) data for the spacecraft and provide command control to (uplink) and data reception from (downlink) the spacecraft. Large antennas, low-noise phase-lock receiving systems, and high-power transmitters are utilized. The 85-ft-diameter antennas have gains of 53 db at 2300 MHz, with a system temperature of 55°K, making possible significant data rates at distances as far as the planet Mars. To improve the data rate and distance capability, a 210-ft-diameter antenna was built at the Mars DSS, and two additional antennas of this size are planned for installation at overseas stations. In their present configuration, all stations except the Johannesburg DSS are full S-band stations. The Johannesburg DSS receiver has the capability for L- to S-band conversion.

It is the policy of the DSN to continuously conduct research and development of new components and systems and to engineer them into the network to maintain a state-of-the-art capability. Therefore, the Goldstone DSCC is also used for extensive investigation of space tracking and telecommunications techniques, establishment of DSIF/spacecraft compatibility, and development of new DSIF hardware and software. New DSIF equipment is installed and tested at the Goldstone DSCC before being accepted for system-wide integration into the DSIF. After acceptance for general use, the equipment is classed as Goldstone Duplicate Standard in order to standardize the design and operation of identical items throughout the system.

B. Ground Communication System

To enable communications between all elements of the DSN, the Ground Communication System (GCS) uses voice, teletype, and high-speed data circuits provided by the worldwide NASA Communications Network between each overseas deep space station, the Cape Kennedy DSS, and the Space Flight Operations Facility (SFOF, described below). The NASA Communications Network is a global network consisting of more than 100,000 route mi and 450,000 circuit mi interconnecting 89 stations, of which 34 are overseas in 18 foreign countries. Entirely operationally oriented, it is comprised of those circuits, terminals, and pieces of switching equipment interconnecting tracking and data acquisition stations with, for example, mission control, project control, and computing centers. Circuits used exclusively for administration purposes are not included.

Voice, teletype, high-speed data, and video circuits between the SFOF and the Goldstone DSCC are provided by a DSN microwave link.

C. Space Flight Operations Facility

During the support of a spacecraft, the entire DSN operation is controlled by the Space Flight Operations Facility (SFOF) at JPL. All spacecraft command, data processing, and data analysis can be accomplished within this facility. Operations control consoles, status and operations displays, computers, and data processing equipment are used for the analysis of spacecraft performance and space science experiments. Communications facilities are used to control space flight operations by generating trajectories and orbits and command and control data from tracking and telemetry data received from the DSIF in near-real time. The telemetry, tracking, command, and station performance data recorded by the DSIF are also reduced at the SFOF into engineering and scientific information for analysis and use by scientific experimenters and spacecraft engineers.

VI. DSIF Development and Operations

THE DEEP SPACE NETWORK

A. Flight Project Support

1. *Surveyor* Project

With the Echo DSS as a transmitter backup, the Pioneer DSS tracked the first pass of *Surveyor II* on September 20. Early in the pass, the midcourse maneuver commands were transmitted to the spacecraft. Following the second-pass acquisition of *Surveyor II* by the Pioneer DSS on September 22, unsuccessful attempts were made to stop the vehicle's tumbling. The mission was terminated less than 24 hr before the scheduled soft landing on the Moon.

2. *Lunar Orbiter* Project

Tracking of *Lunar Orbiter I* was performed daily by the Echo DSS until September 16. Previously, on September 14, the Goldstone calibration leader in the spacecraft camera equipment was read out in a final video sequence for a TV system degradation determination. Following the pass on September 16, tracking of the spacecraft was conducted on an extended-mission as-scheduled basis. Ranging and time-correlation experiments were performed between the Echo DSS and the Robledo DSS and the Echo DSS and the Tidbinbilla DSS. Lunar mapping experiments were also conducted.

3. *Pioneer* Project

The *Pioneer VI* spacecraft is currently being tracked by the Mars DSS on a scheduled basis. The Echo DSS provides telemetry and command processing by means of microwave to the Mars DSS. Routine data are being returned from the spacecraft, and the vehicle provides an accurate source for antenna and system testing in progress at the Mars DSS.

Due to the Echo DSS commitment to track *Lunar Orbiter I*, the *Pioneer VII* spacecraft was tracked by the Pioneer DSS from August 17 (launch) until September 7. Telemetry and command processing from the *Pioneer* ground operations equipment and the telemetry and command processing beta computer was provided by the Echo DSS to the Pioneer DSS On September 9, the Echo DSS assumed prime tracking on a daily basis until October 1. Extended-mission, as-scheduled tracking is currently being performed 3 to 4 days each week.

4. *Mariner* Mars 1964 Project

The *Mariner IV* spacecraft continues to be tracked jointly by the Mars DSS and the Venus DSS. During a scheduled monthly track, two-way lockup is established, commands are transmitted, and telemetry is received and recorded. The Mars DSS performs the tracking, necessary command transmissions, and receiving, and the data are microwaved to the Venus DSS for processing. During October, a series of low-level telemetry experiments was performed.

5. *Mariner* Venus 67 Project

Although the ground instrumentation for the *Mariner* Venus 67 mission will be similar to that used for the *Mariner IV* mission, there will be marked differences due to the very dense atmosphere of Venus. One effect the atmosphere will have is that the doppler perturbations are expected to be of the order of 30 kHz (in contrast to 3 Hz, one way, in the case of *Mariner IV*). In addition, the atmosphere will act as a defocusing lens which produces refractive attenuation estimated to be as high as 26 db. To ensure the most favorable possible signal-to-noise ratio, the Mars DSS with its 210-ft-diameter antenna has been chosen as the prime station for the *Mariner* Venus 67 occultation experiment. The preliminary configuration employs three separate receivers: (1) a standard phase-lock-loop DSIF configuration to be used for telemetry and doppler data acquisition, (2) a slightly modified standard receiver with a fixed tuned local oscillator derived from the frequency standard by use of a synthesizer, and (3) an alidade instrumentation area receiver with a programmable local oscillator for the occultation experiment. During the experiment, the Pioneer DSS will receive doppler data in the normal configuration, and the Venus DSS will use an open-loop receiver.

Ranging was approved as a part of the *Mariner* Venus 67 celestial mechanics experiment. The experiment will require the Mars DSS 210-ft-diameter antenna fitted with a 20-kw transmitter, plus a supplement to the existing receiver (discussed in Section C). It is probable that at least 1 to 1.5 hr per pass at the Mars DSS will be necessary during the pre-encounter period. Measurements immediately prior to encounter and as near post-encounter emergence as feasible will be of extreme value in determining the astronomical unit.

B. DSS Equipment Installation

1. Pioneer DDS

Installation and system testing of the MSFN equipment are progressing. The RF switching box, enabling rapid transfer of the antenna and antenna-associated equipment, was installed and operationally tested. The tower and antenna for the microwave link between the Pioneer DSS and the Apollo Station are operational.

The waveguides and the RF power combiner for the dual transmitter on the antenna are being installed. Equipment of the S-band system requiring certification was routed through the Goldstone calibration, certification, and maintenance facility, following termination of the *Surveyor I* mission. The S-band receiver modules, many of which are the original pilot modules, were calibrated and certified to establish operational levels comparable with those of the latest S-band systems. Concurrently, relocation of racks at the multiple mission support area continued in order to provide space for the new equipment.

2. Echo DSS

The multiple mission support area/mission support recording (MMSA/MSR) equipment has been installed and operationally tested. The rack rearrangement of the microwave, recording, and analog instrumentation subsystems to provide area for the installation of the MMSA/MSR equipment also necessitated relocation of the communications teletype equipment.

3. Venus DSS

An enclosure to contain moving portions of the cable wrapup was added to the 30-ft-diameter azimuth–elevation antenna. To reduce structural deformation due to thermal gradients, the complete antenna structure, with the exception of the surface of the dish, was painted white. The 85-ft antenna was also completely repainted to minimize thermal gradients.

To provide dual-frequency receiving capability in one antenna cone, the 2295- to 30-MHz converter and power supply were removed from the *Mariner* cone and installed in the 2388-MHz research-and-development cone. Appropriate cabling was also installed, thus providing the two converters with independent control and signal functions. A 400-cycle power cable change is the only modification required to switch reception from one frequency to the other.

Due to continuing tape-reader problems in the digital portion of the programmed local oscillator, a complete mechanical and electrical overhaul of the tape reader and spooler was performed.

4. Mars DSS

Except for the maser and associated equipment not presently in use, installation and operational testing of the S-band system are nearing completion. Subsystem operational testing is being conducted during *Pioneer VI* tracking.

C. Communications Development and Testing

1. S-Band Receiver–Exciter Subsystem

The original S-band receiver–exciter subsystem, designated Block I, was designed specifically for DSN requirements. This subsystem used with the Mark I ranging subsystem can precisely determine ranges up to 800,000 km when used in conjunction with an S-band "turnaround" transponder. However, use of this S-band receiver–exciter subsystem for the MSFN has necessitated the addition of certain new designs, as well as changes in some existing designs. The receiver–exciter subsystem in which these design changes will be incorporated has been designated Block II. Further design improvements have been necessitated by the DSN and MSFN requirements for switching from one channel to another without having to rephase and recalibrate the ranging receiver and for a subsystem capable of supporting both DSN and MSFN tracking commitments. The new designs to meet these two requirements are designated Blocks IIB and IIIC, respectively.

a. Block IIB design. The Block II subsystem uses a range code synchronizing signal (clock) derived from the exciter voltage-controlled oscillator. This is mechanized by multiplying the voltage-controlled oscillator frequency by a factor of 5/221. The clock is also used as a reference signal for the ranging receiver and transfer loop. This configuration makes it necessary to recalibrate the ranging receiver each time a large frequency change is made. To eliminate the need for this recalibration, the Block IIB design uses a fixed clock and reference frequency derived from the station standard. This frequency establishes a relatively constant delay in the ranging receiver. A fixed clock frequency of 496 kHz was selected to be compatible with most of the existing hardware. The clock signal is derived by multiplying a 1-MHz signal output from the tracking station standard by a factor of 496/1000.

To obtain a completely common system, which is the purpose of the Block IIIC design, the three Block II DSN loop noise bandwidths are used for both the Block IIB DSN and MSFN receivers. The DSN noise bandwidths (0.8, 4, and 12 Hz) are adequate to meet the requirements of both the DSN and MSFN.

The fixed clock frequency causes a non-integer ratio to exist between the clock and RF doppler frequency. The doppler counting logic of the Mark I ranging subsystem must be changed to allow the RF doppler to be tallied correctly.

b. Block IIIC design. The existing Block II design does not provide the capability of tracking both DSN and MSFN space probes in the following areas: (1) exciter tuning range, (2) receiver tuning range, (3) reference receiver predetection and loop noise bandwidths, (4) receiver automatic-gain-control loop noise bandwidth, and (5) ranging receiver predetection and loop noise bandwidths. Regarding the exciter tuning range, the $\times 32$ frequency multiplier of Block II has been replaced by a design having adequate bandwidth to cover both the DSN and MSFN bands. This is accomplished in two subassemblies: a $\times 4$ frequency multiplier and a $\times 8$ frequency multiplier. The +27.5-dbm minimum output level of the Block IIIC multiplier chain provides adequate drive to obtain the required 20-MHz bandwidth (for the combined DSN and MSFN 10-MHz bands) from the existing Block II UHF buffer amplifier. The receiver local oscillator multiplier chain of Block IIIC will use the same design as that of the exciter.

To provide the predetection filter bandwidths, a new 10-MHz filter amplifier subassembly was added to the subsystem ahead of the 10-MHz IF amplifier. The receiver loop filter module contains the time constants for both the three DSN and the three MSFN loop noise bandwidths. Three receiver automatic-gain-control loop time constants which adequately cover both the DSN and MSFN requirements have been selected. These bandwidths are common to both systems, and no selection is is required. As in the automatic-gain-control loop, a design common to both the DSN and MSFN systems has been selected for the range receiver.

c. Summary. Blocks IIB and IIIC provide a system with both the DSN and MSFN parameters and the capability to select these parameters by switch control. The parameter differences have now been reduced to the reference receiver predetection and loop noise bandwidths. However, the exciter and receiver voltage-controlled oscillator subassemblies still contain only four selectable channels. Two sets of voltage-controlled oscillator assemblies are provided in Block IIIC, one for the four MSFN channels and one for four DSN channels. Switching from an MSFN channel to a DSN channel or vice versa requires that the voltage-controlled oscillator subassemblies installed in the subsystem be replaced with the second set.

Present plans are to upgrade all DSN and MSFN receiver–exciter subsystems to the Block II, IIB, and IIIC designs.

2. Traveling Wave Maser for Use at 8448 MHz

A traveling wave maser (TWM) has been constructed for use at 8448 MHz. The structure accepts both signal and pump waveguides at one end. Transitions from rectangular to ridged waveguides are used to reduce the maser diameter. The slow-wave comb structure is folded, resulting in 4 in. of active length in 2 in. of uniform magnetic field. All parts are oxygen-free high-conductivity copper, and pure indium sheeting is placed in all joints to prevent RF leakage and to improve heat transfer across these joints. Fig. 1 shows the maser structure installed in a closed-cycle helium refrigerator developed at JPL.

The net gain of the maser (41.5 db when installed in a closed-cycle helium refrigerator operating at 4.4°K) has a short-term stability (min) better than 0.1 db, but the long-term stability is seriously impaired by temperature variations affecting the magnet. The measured equivalent input noise temperature of 18°K is higher than the predicted value (based on insertion loss measurements of input waveguide components and the calculated maser noise temperature). The maser can be tuned over a 140-MHz range with the net gain exceeding 30 db. Cooldown time of the closed-cycle helium refrigerator with this maser installed is 4.6 hr with no liquid-nitrogen precooling and 2.4 hr with precooling.

During laboratory testing, maser gain changes occurred when changing input terminations. These changes were found to be a function of reflected pump energy, which behaves in an unpredictable manner in the signal waveguides. Gain changes as large as 1 db occurred, even though the maser appeared well-saturated with pump power. A low-pass waveguide filter (15-GHz cutoff frequency) was installed at the maser input. This filter has $>$ 40-db rejection at the pump frequency and 0.042-db insertion loss at 8448 MHz. Gain changes due to input load changes are now less than can be measured with existing test equipment.

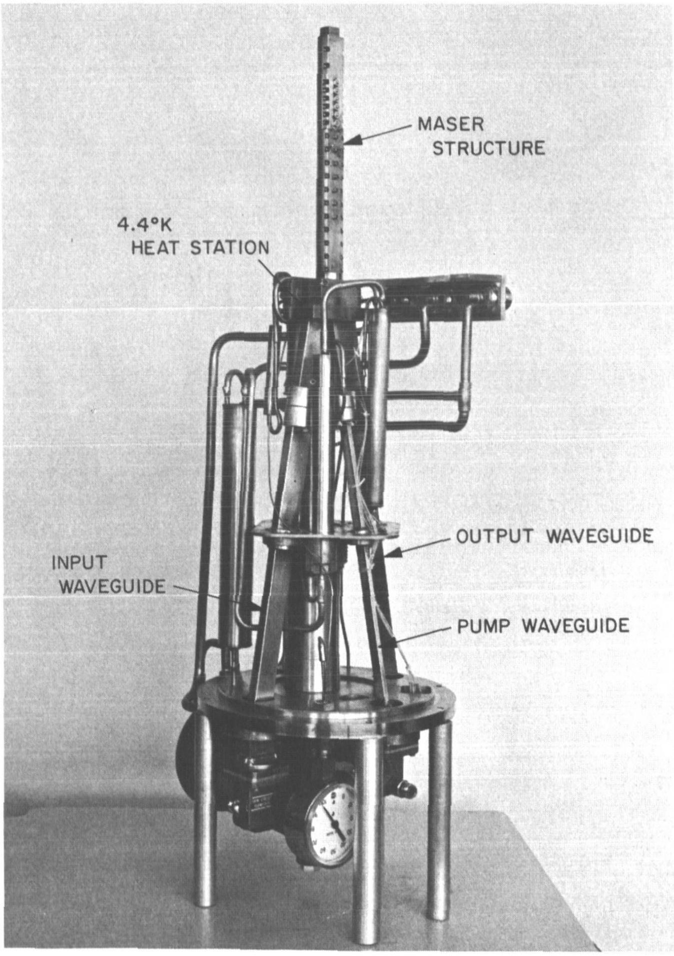

Fig. 1. Maser installed in closed-cycle refrigerator

3. X-Band Cone RF Instrumentation

The X-band cone is immediately adaptable to the Venus DSS and Mars DSS antennas and will be used to provide operational performance data for these antennas at 8448 MHz. A total power radiometer, using a maser amplifier and precision waveguide terminations, will be used for the necessary noise temperature calibrations. The first installation of the experimental cone will be at the Venus DSS. The initial phase of component installation is shown in Fig. 2.

Preliminary noise temperature calibrations of the receiving system were made using the horn, maser, and monitor receiver. The equipment was used in a total power radiometer configuration with precision cryogenic waveguide terminations. When the results are available, the difference between the system temperature measured with the horn connected directly to the maser input and that measured in the cone with the normal waveguide installation will provide a measure of transmission line losses and a cross check on the normal waveguide insertion loss measurements.

4. *Mariner* Venus 67 Ranging Equipment

The research-and-development receiver ranging supplement at the Mars DSS to be used for *Mariner* Venus

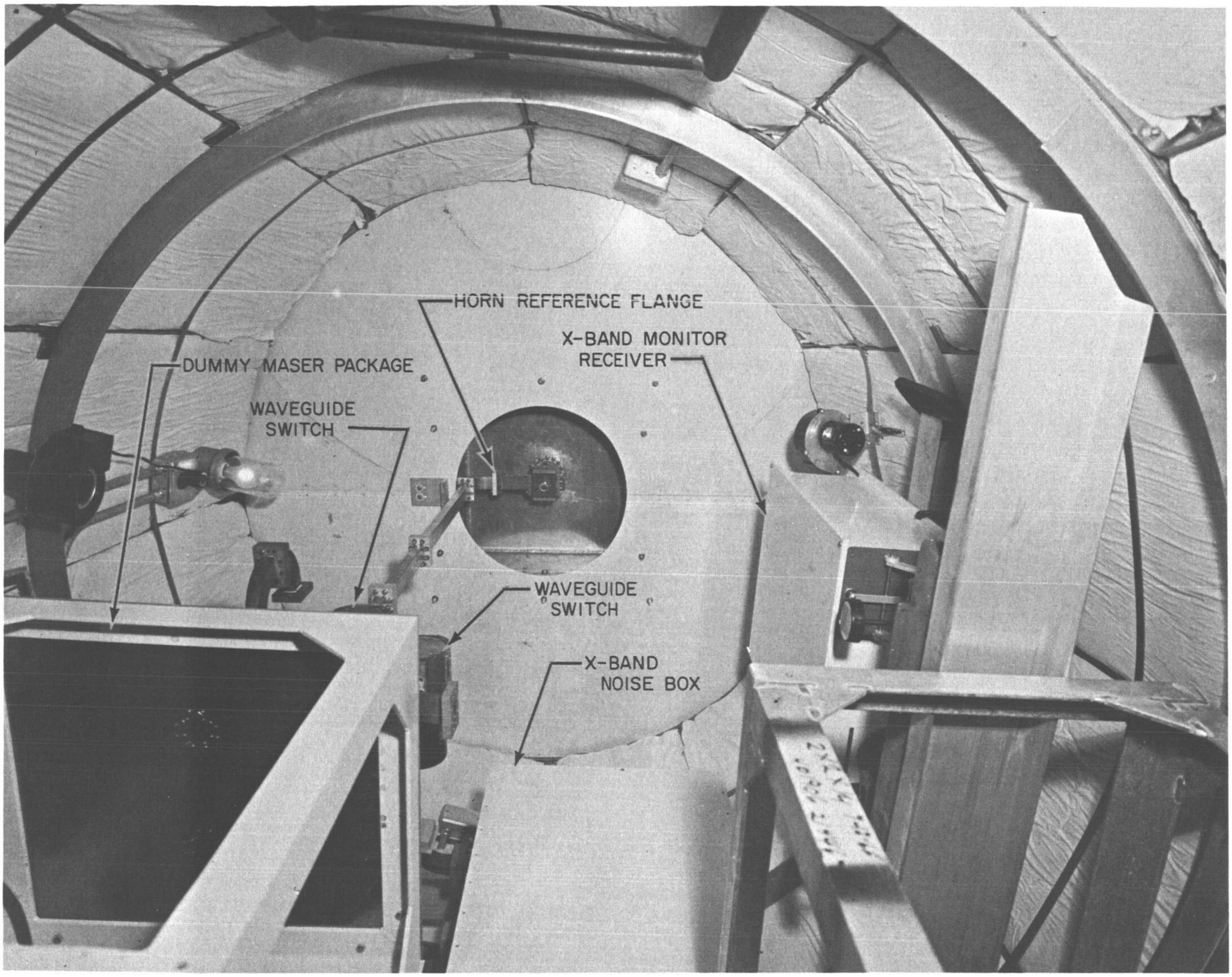

Fig. 2. X-band cone interior during initial phase of component installation

67 range measurements consists of two correlation channels and a stored program signal processor. The correlation channels are constructed from modules very similar or identical to modules in the Goldstone Duplicate Standard receivers. To provide the IF stability required for the narrow bandwidths which will be involved, the usual free-running reference oscillator in the Goldstone Duplicate Standard receiver is replaced by an atomically derived reference.

The signal processor consists of a Scientific Data Systems (SDS) 920 general-purpose computer with analog-to-digital and digital-to-analog interfaces, supplemented by a number-controlled oscillator and various digital phase detectors, counters, and other components. The automatic feature of the programmed devices makes it possible to effect code acquisition and range tracking in the most effective way; e.g., minute gain offsets in each channel can be compensated for by switching codes and channels back and forth periodically.

The transmitted ranging code will be a six-component composite code of the same general type used in the Mark I ranging system. The receiver component codes will be duplicates of those transmitted and will be generated at a rate set by a voltage-controlled oscillator programmed to track the incoming code rate.